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FATIGUE AND RESIDUAL STRENGTH INVESTIGATION OF ARALL®-3 AND GLARE®-2 PANELS WITH BONDED STRINGERS

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SUMMARY

Stiffened panels were fabricated from ARALL-3 and GLARE-2 laminates for the purpose of providing improved structural performance of lower wing panels for aircraft. To verify the designs fatigue crack growth and residual strength tests were conducted and compared to those for conventional monolithic aluminum panels.

INTRODUCTION

The development of a new aircraft is significantly influenced by the introduction of new structural materials. The driving force behind such development is the everlasting incentive to reduce aircraft operating costs. Future operating costs will incur even greater fuel and maintenance costs, making it cost effective for aircraft structures to be designed with new materials to obtain a lower weight and to require less maintenance. Maintenance is directly related to durability and damage tolerance. ARALL (Aramid Fiber Aluminum Laminate) and GLARE (Glass Fiber Aluminum Laminate) laminates, illustrated in Figure 1, were designed to combine the high static strength of fibers with good crack growth resistance and impact tolerance of aluminum alloys. The excellent fatigue strength of the laminates is unique due to the bridging effect of the crack tip fibers in the prepreg layer of the laminates.

ARALL and GLARE laminates were originally developed by Delft University. They incorporate adhesively bonded laminates which are built up as laminated sheet materials with thin high strength aluminum alloy sheets and strong unidirectional or woven aramid or glass fibers, impregnated with a thermoset or thermoplastic resin followed by (if desired) post-stretch of the material after curing, which results in a compressive residual stress in the metal sheets. ARALL and GLARE laminates are a new family of structural composite materials. The final properties are highly dependent on the variables of the material. They can be tailored for many different applications by varying fiber-resin systems, aluminum alloys, sheet gauges, stacking sequences, fiber orientations (such as uniaxial and cross-ply), surface preparation techniques, and by the degree of post-cure stretching and rolling. Several ARALL and GLARE laminates are listed in Table 1.

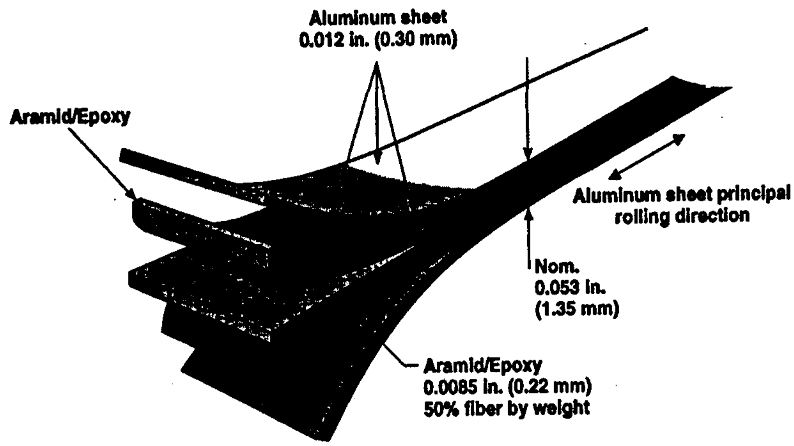


Figure 1: 3/2 ARALL-1 Laminate Layup.

Table 1: ARALL and GLARE Laminates [1]

Laminates	Alloy	Composition	Orientation	Post stretched
ARALL-1	7075-T6	Aramid/epoxy	U.D.*	None
ARALL-2	2024-T3	Aramid/epoxy	U.D.	None
ARALL-3	7475-T76	Aramid/epoxy	U.D.	0.4%
ARALL-4	2024-T8	Aramid/epoxy	U.D.	None
GLARE-1	7075-T6	Glass/epoxy	U.D.	0.4%
GLARE-2	2024-T3	Glass/epoxy	U.D.	None
GLARE-3	2024-T3	Glass/epoxy	Cross ply (50/50)	None
GLARE-4	2024-T3	Glass/epoxy	Cross ply (70/30)	None

* U.D. stands for unidirectional.

Developing a new aircraft structure is a complicated and lengthy process due to the high safety requirements of the airplane. Every aspect of the new material must be fully tested and analyzed. Fatigue and residual strength are two of the most important aspects of damage tolerance evaluation of the material. The primary objectives of this investigation were to build the stiffened center notched panel specimens using ARALL and GLARE laminates based on the existing lower wing cover design, conduct fatigue crack growth tests under constant amplitude loading, conduct residual tension strength tests, and compare the test results with those from the conventional 2024 aluminum structures.

SPECIMEN DESIGN AND EXPERIMENTAL PROCEDURES

Test Specimens

Two stiffened test panels (one of ARALL-3 and one of GLARE-2) were designed based on a 2024-T351 panel that was tested during Gulfstream Aerospace's G-IV wing program. The panels were fabricated at Textron Aerostructures. The original ARALL-3 and GLARE-2 sheets were manufactured by Structural Laminates Company of ALCOA. The cross-sectional area of the panel was kept the same to enable a direct comparison of these two different types of materials. The panels were 28.0 inches wide and 75.9 inches long with three Z-shaped stringers and two simulated beam caps bonded to the composite skins. A sketch of the fatigue panel is shown in Figure 2 (The crack size shown in the Figure is not the actual size). The skin pads were used between the stringers and the panels. The 1.4 inch initial crack, $2a$, was induced by sawcut. The sawcut was through the middle stringer and skin. The cut plane was perpendicular to the skin surface within $\pm 2^o$ (Figure 3).

Z-shaped stringers, made of 7075-T6511 aluminum extrusions, were bonded to ARALL and GLARE panels with AF163 film adhesive. The ARALL-3 and GLARE-2 panels (anodized at Structural Laminates Company) were primed with corrosion inhibiting primer and then dried in oven at $250^o F$. The stringers were first anodized and primed like the panels. The stringers and panels were then vacuum bagged and cured in an autoclave with 40 psi pressure at $250^o F$ for 90 minutes to achieve the optimum bonding. The material properties used in this investigation are tabulated in Table 2 [2].

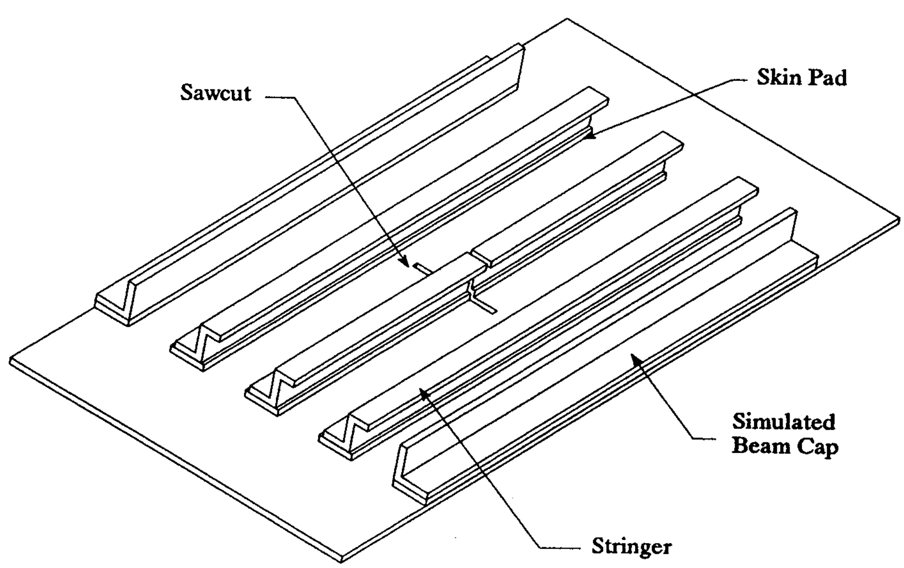
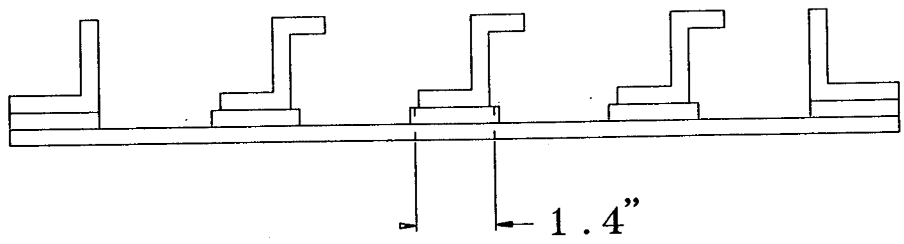


Figure 2: Design of Stiffened Center Notched Laminate Panel.



(Initial Crack Length)

Figure 3: Sawcut Location and Size for Fatigue Tests.

Test System and Experimental Procedures

The testing of these two panels was conducted at Alcoa Technical Center. An MTS servo-hydraulic testing machine with 250 *kips* dynamic loading and 400 *kips* static loading capacity was used to apply the cyclic and static tension load to the panels. The test system is shown in Figure 4.

The test panel was first mounted on the testing machine with upper and lower ends clamped by the gripping fixture as shown in Figure 4. A constant amplitude load spectrum of 10.0 *kips* to 47.3 *kips* was applied to both test panels for 85000 cycles at 5 *Hz* frequency. The crack length measurements were recorded every 5000 cycles with optical magnifying device.

The cracks were sawcut to 9.73 inches (dimension between first pair of out stringers) after 85000 cycles of fatigue crack growth tests (Figure 5). This crack length was to be the initial crack extension for the residual strength tests. Since the fatigue crack growth for the ARALL-3 and GLARE-2 stiffened panels was much lower than that for the 2024 stiffened panel due to the excellent fatigue resistance of these fiber/metal laminates, sawcut was introduced to extend the initial crack to the same length as for the 2024 stiffened panel. Both ARALL-3 and GLARE-2 test panels were then loaded till fracture in the same testing machine.

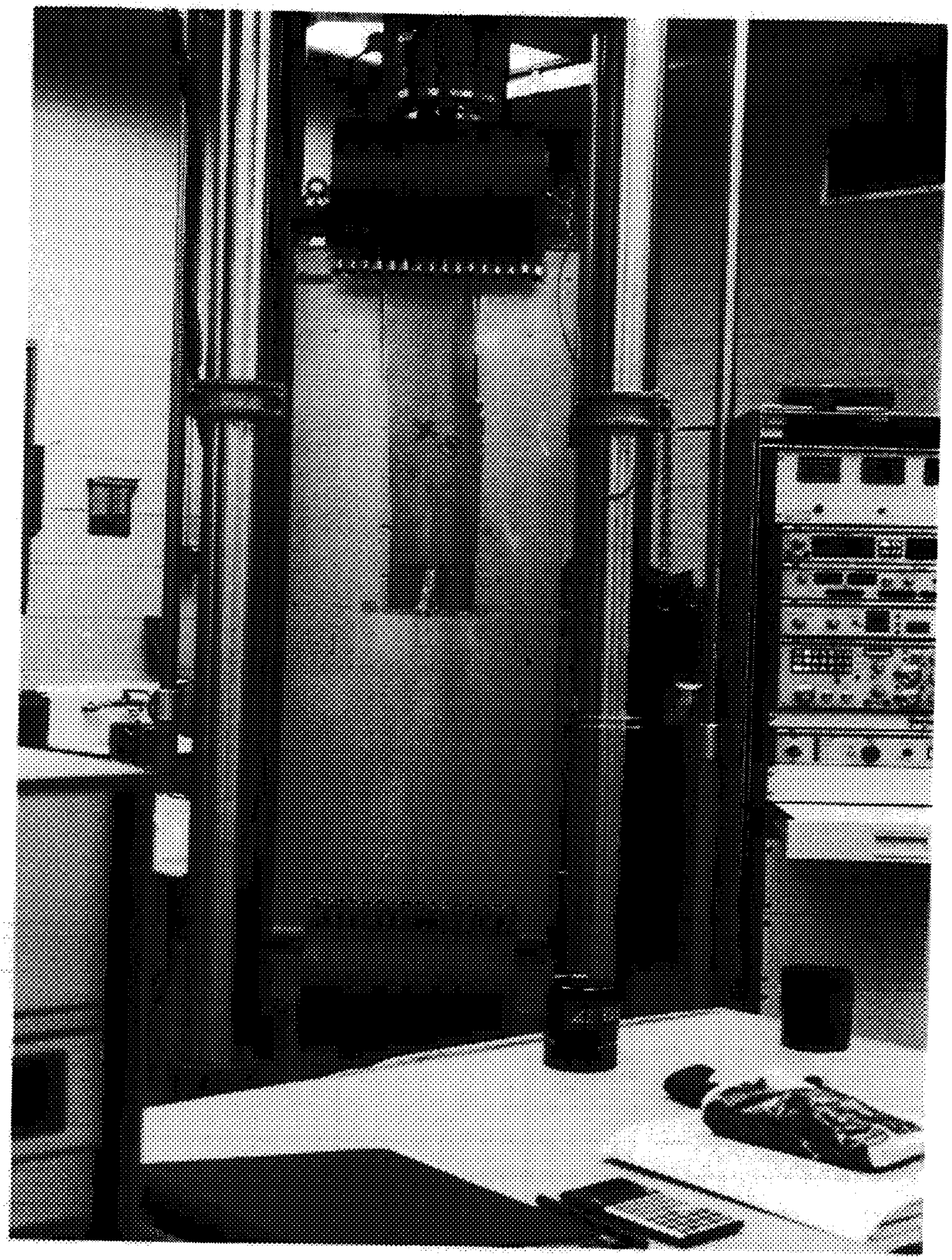


Figure 4: Test System Set-Up.

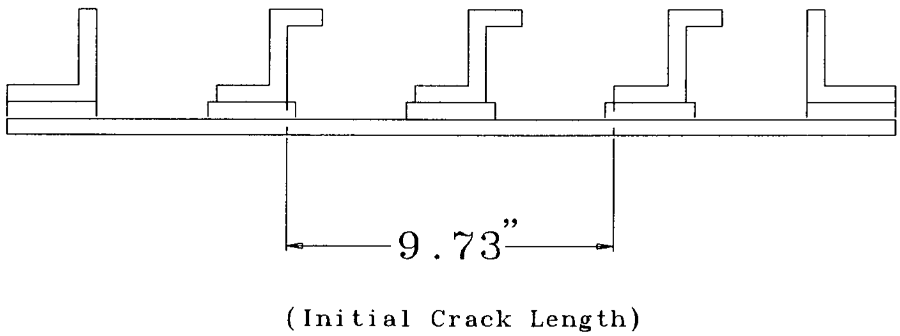


Figure 5: Sawcut Location and Size for Residual Strength Tests.

Table 2: Mechanical Properties of Materials

		STRINGER	UPPER SKIN		LOWER SKIN		
		7075 T6511 extr QQ-A-200/11 0.750"-1.490"	7075 T7351 plate QQ-A-250/12 0.500"-1.000"	2090 T83 sheet AMS 4251 0.126"-0.249"	2024 T351 plate QQ-A-250/4 0.500"-1.000"	ARALL-3 5/4 sheet AMS 4302A 0.094"	GLARE-2 4/3(5/4) sheet 0.078"(0.100")
F_{tu}, ksi	L	94.7	70	84.8	65	123.9	173.3
F_{ty}, ksi	L	84.7	59	79.2	50	89.7	56.1
F_{cy}, ksi	L	87.7	58	71.3	41	47	59.1
F_{su}, ksi		49	39	41.9	38	34.2	-
F_{bru}, ksi e/D=1.5 e/D=2.0		 122 152	 106 136	 100 126	 98 120	 82 84	 - 103
E_t, msi	L	10.4	10.3	11.3	10.7	9.2	9.4
E_c, msi	L	10.7	10.6	10.6	10.9	9.2	9.4
G, msi		4	3.9	-	4	2.3	-
μ		0.33	0.33	-	0.33	0.35	-
n		26	20	20	9	5.3	-
w, lb/in^3		0.101	0.101	0.093	0.1	0.081	0.09
t, in		-	-	-	-	0.094	0.078(0.100)
cost, \$/lb		2.5	4	15	3	58	56

TEST RESULTS AND DISCUSSIONS

The fatigue crack growth tests for both stiffened center notched ARALL-3 and GLARE-2 panels show considerable reduction of crack growth rate compared to 2024 aluminum stiffened panels. The test results are tabulated in Table 3. For ARALL-3 panel, $P_{max} = 51$ *kips*, $P_{min} = 11$ *kips*, and frequency = 7 *Hz*. For GLARE-2 panel, $P_{max} = 48$ *kips*, $P_{min} = 10$ *kips*, and frequency = 5 *Hz*. The comparison of fatigue crack growth curves between stiffened ARALL-3 and GLARE-2 laminated panels and stiffened 2024 aluminum panel is shown in Figure 6.

Table 3: Fatigue Crack Growth Test Results for ARALL-3 and GLARE-2 Stiffened Panels

	ARALL-3		GLARE-2	
No. of Cycles	a, left	a, right	a, left	a, right
0	0.70	0.70	0.70	0.70
5000	0.73	0.72	0.78	0.74
10000	0.75	0.74	0.84	0.79
15000	0.76	0.75	0.87	0.82
20000	0.78	0.76	0.90	0.83
25000	0.79	0.76	0.92	0.84
30000	0.80	0.78	0.95	0.86
35000	0.81	0.78	0.97	0.88
40000	0.82	0.78	0.99	0.89
45000	0.83	0.79	1.01	0.90
50000	0.84	0.80	1.02	0.91
55000	0.84	0.80	1.04	0.92
60000	0.84	0.81	1.06	0.92
65000	0.85	0.82	1.07	0.94
70000	0.86	0.82	1.08	0.94
75000	0.87	0.82	1.09	0.94
80000	0.87	0.82	1.10	0.96
85000	0.88	0.83	1.11	0.96

The higher fatigue crack growth resistance of ARALL-3 and GLARE-2 laminate panels is due to the bridging mechanism of fibers at the crack tips which prevents the crack from further opening. The cyclic crack closing fiber stresses, due to crack bridging, are partly transferred into the aluminum across the adhesive. This causes delamination in the adhesive behind the crack tip due to cyclic shear loading of the adhesive interface between the fibers and the aluminum sheets. If this delamination is assumed to be absent and the adhesive exhibits an infinite shear modulus, the crack bridging of the fibers will be perfect. The crack can not open. Consequently the stress intensity factor at the crack tip is zero,

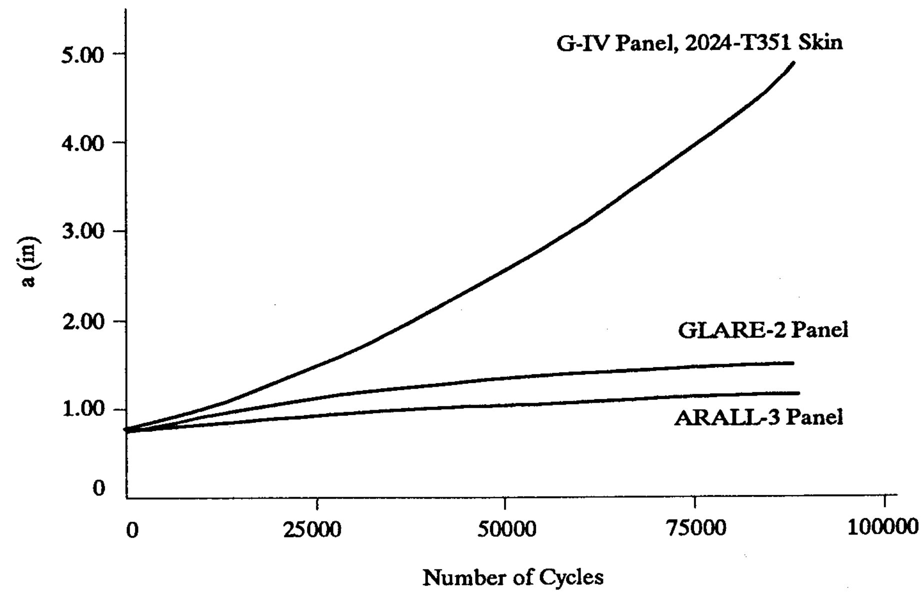


Figure 6: Crack Extensions vs Number of Cycles.

and the fatigue crack growth will not occur. The load transmission "through" the crack occurs by the fibers only. Due to the stress concentration caused by the notch, the fiber loads will become high, especially at the end of the starter notch. This could cause fiber failure, which implies a complete loss of crack bridging effect. Therefore, a perfectly stiff adhesive between the layers and a zero delamination are not an optimum condition for the ARALL and GLARE laminates. In reality, there is some adhesive shear deformation which allows some crack opening. Further, some delamination will occur around the crack, especially at locations where the fiber stresses are high. As a consequence, a redistribution of the fiber stresses along the crack occurs. Because the redistribution of the fiber stress along the crack implies a much lower peak load value at the notch root, fiber failure behind the crack tip can be avoided. However, the amount of adhesive shear deformation and delamination should remain within certain limits. Otherwise, the crack bridging efficiency would become too small. Fortunately, for ARALL and GLARE laminates, these aspects are not critical.

The crack growth rates of ARALL-3 and GLARE-2 panels decreased with increasing crack length as demonstrated in Figure 6. This is due to the increasing amount of crack bridging fibers with increasing crack length.

The residual strength test of both ARALL-3 and GLARE-2 stiffened panels demonstrated an increase of strength over 2024 aluminum stiffened panel with the same initial crack size (9.73 inches) as shown in Figure 7. This is due to the high tensile strength aramid or glass fibers in these laminates. Buckling and bending were observed during the residual strength tests. The buckling was caused by compressive stress in the direction perpendicular to the loading direction and the bending was due to the off-centroid loading

of the panels. Severe buckling around the initial crack was the reason that shearing occurred at the crack tip. The gripping fixture used for these tests was smaller than the panel width. This was the other reason that caused the shearing, which could have affected the true strength results of these laminate panels. The failed ARALL-3 and GLARE-2 panels are shown in Figures 8 to 11.

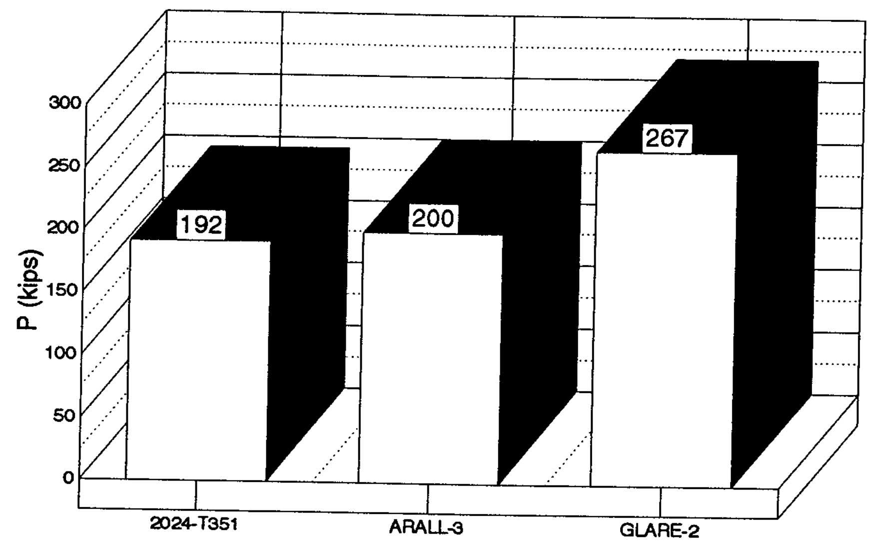


Figure 7: Residual Strength Test Results for a 9.73 in. Initial Crack Length.

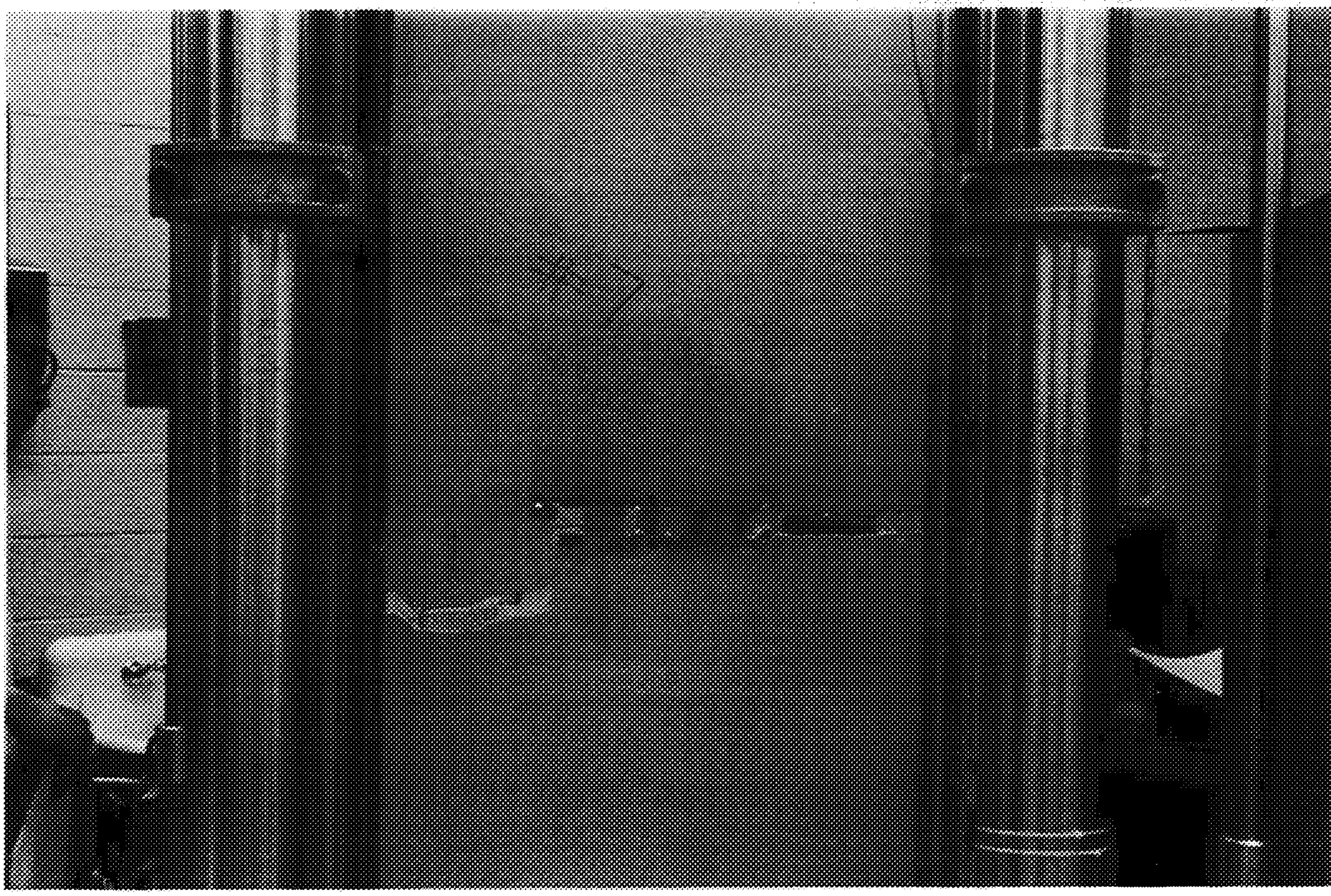


Figure 8: Skin Side of Failed ARALL-3 Stiffened Panel.

Figure 9: Stringer Side of Failed ARALL-3 Stiffened Panel.

Figure 10: Skin Side of Failed GLARE-2 Stiffened Panel.

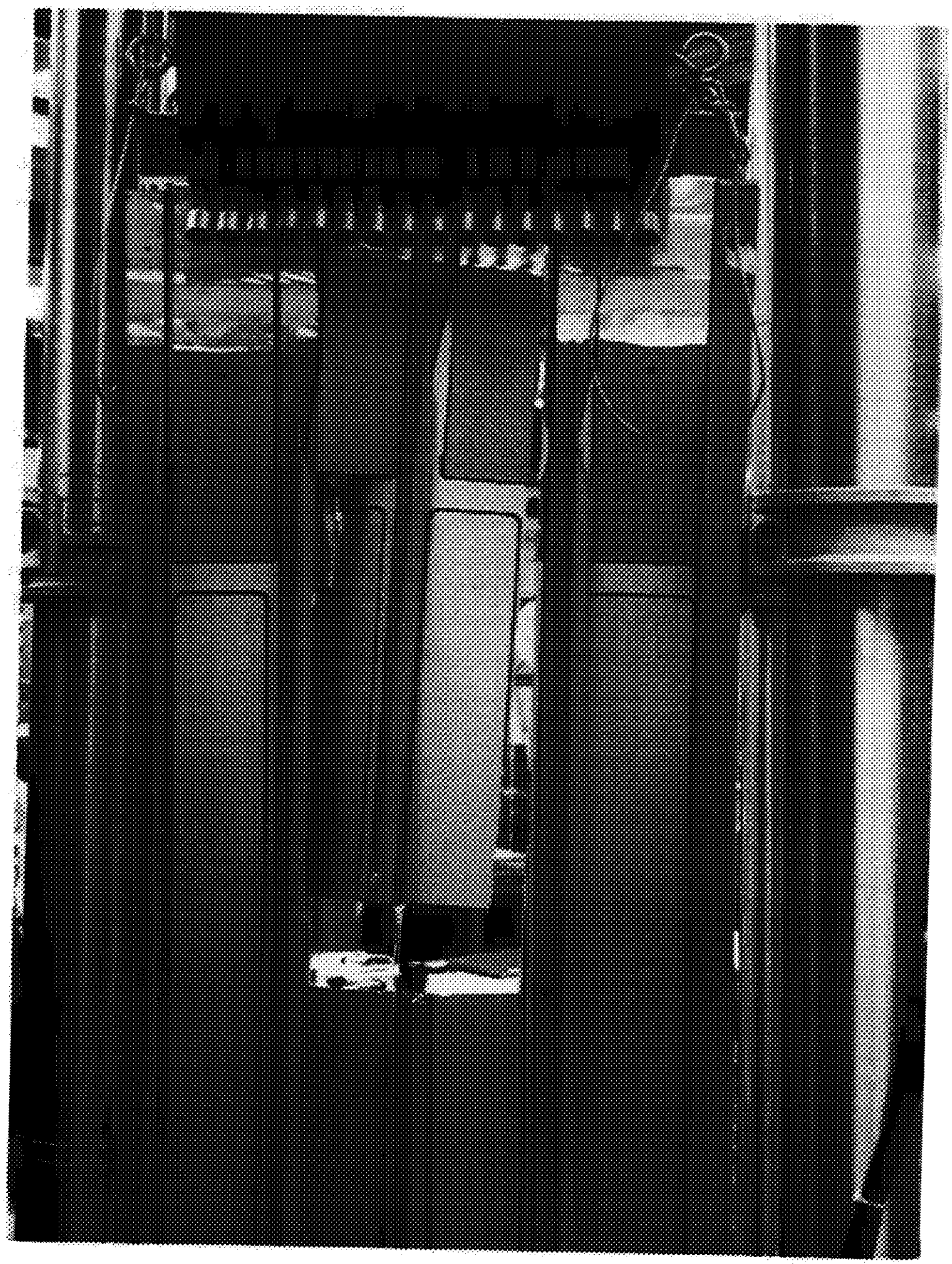


Figure 11: Stringer Side of Failed GLARE-2 Stiffened Panel.

CONCLUSIONS

The fatigue crack growth tests showed that the ARALL-3 and GLARE-2 panels have a much lower crack growth rate than that of an equivalent 2024 stiffened panel which was tested for the G-IV wing program. These laminated panels were sized to have the same cross sectional areas and initial crack lengths as the 2024 panel. The residual strength of ARALL-3 and GLARE-2 panels was higher than that of the 2024 panel due to the high strength aramid and glass fibers in the laminates. The improvement of fatigue and residual strength over aluminum, combined with lower density, gives these fiber/metal laminates a potential application in future aircraft structures.

REFERENCES

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2. Ming Wu, Reddy, S. V., and Dale Wilson, "Design and Testing of Z-Shaped Stringer Stiffened Panels – Evaluation of ARALL, GLARE, and 2090 Materials," Proceedings, ASM/TMS Material's Week, to be Published, October, 1994.